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# 1. INTRODUCTORY CONCEPTS

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## THE CHOICE

It is clear that the military planners today face some difficult and far-reaching decisions concerning the choice of deterrent weapons to be developed for the future. These weapons systems include the manned bomber and the unmanned missile for sustained flight within the atmosphere; the glide bomber; and, beyond the atmosphere, the intercontinental ballistic missile and the satellite bomber. All these systems have their chemical and nuclear counterparts. Although each has its own virtues, only the ICBM has been assured of vigorous support at the moment of this writing.

This situation is at least in part due to the fact that these various weapons systems have a common vice. They are all expensive and time-consuming to develop. This does not mean that only the least expensive system should be developed, however, or that only one should be developed. Since each unit is capable of such vast destruction, fewer units are needed. Therefore, the choice may be made on the basis of criteria other than cost. It is probable, however, that all these systems cannot be developed simultaneously.

One of the most tensely awaited outcomes of this deliberation will be the role of the air-breathing engine. Most of the aircraft industry has been developed around this type of engine and the airframe it powers. Before a rational decision can be made, however, a vast amount of information must be gathered about the various weapons systems. This is the purpose of these first five papers - to contribute to this fund of information by presenting an appraisal of the ultimate performance capabilities of aircraft and missiles powered by air-breathing engines.

## CRITERIA OF MERIT

There are many criteria of merit to be considered in evaluating any type of weapons system. Some of the more important are range, speed, weight, payload, accuracy, reliability, vulnerability, development time, useful life, cost, flexibility, and logistics. Of these, only range, speed, weight, and payload have been evaluated. The other criteria, with

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the exception of development time, are beyond the scope of this study. In this regard it seems probable that ten years would be required to develop an aircraft or missile utilizing the powerplants discussed herein.

## REGIONS OF SUSTAINED FLIGHT

The probable regions of future sustained flight within the atmosphere are presented in figure 1. Today none of our subsonic manned aircraft has an unrefueled radius approaching the 5500-mile target distance established by the military some years ago (the 6500- and 8500-mile marks in the figure are hypothetical future goals considered in paper 5 on Mission Studies). Admittedly, the manned aircraft can extend the useful radius by aerial refueling, "fly-over" missions, and, from a deterrent point of view, could even be considered for their one-way capability. The unmanned Snark, however, attains the 5500-mile range, since its missions are all one way.

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The supersonic bomber, the B-58, utilizes a split-speed mission to achieve a fairly limited unrefueled radius. The currently proposed second generation of supersonic bombers, the WS-110, are designed to cruise at Mach 3 over ranges approaching those of our current subsonic bombers. Still longer ranges are certainly desirable, and again the one-way missile can achieve them, as typified by the now defunct advanced version of the Navaho. This missile represented the only ramjet-powered bombardment vehicle.

The WS-110 and the advanced Navaho probably represent about the limits to which present technology can be pushed. The question is whether additional research and development can yield appreciably better performance for both the piloted bomber and the unmanned missile. Examination of figure 1 indicates that the most obvious need of the manned bomber is greater range capability. If missile performance is to advance appreciably beyond that projected for the Navaho, flight at very high stagnation temperature will be necessary.

## COOLING

The temperature problems of high-speed flight are visualized in figure 2 where various skin temperatures are plotted as functions of flight Mach number. Also indicated are some assumed materials limits for combustor and other surfaces.

Radiation cooling at the high altitudes accompanying high speeds is sufficient to maintain the external surfaces at marginally acceptable levels. Unfortunately, the interior passages cannot radiate. Above Mach 4.5 the subsonic diffuser temperature exceeds the materials limits and,

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hence, these surfaces must be cooled. Since this temperature would also apply to the compressor of turbojet-type engines, and since cooled compressors are not foreseeable, Mach 4.5 probably represents the absolute upper limit for this type engine. Actually, an upper limit closer to Mach 4 is probably more reasonable, and even at this speed the lubricants must be cooled.

For the range of flight speeds where the diffuser temperatures are well below combustor temperatures, film cooling can be used to minimize fuel-cooling requirements even though the temperature of the cooling film of inlet air actually exceeds materials limits.



## FUELS AS COOLANTS

Because the concept of cooling with the fuel as it flows from the tank to the combustor has been introduced, the adequacy of such a source of cooling should be discussed. JP fuels and ethyldecaborane break down if they are permitted to heat up much. Cryogenic fuels like diborane, liquefied methane, and liquefied hydrogen cannot be maintained as liquids if their temperatures are allowed to rise. However, since they are burned as gases, this is not particularly worrisome provided that any phase change occurs before the cooling passages and that the resulting gas has a reasonably high specific heat and can be heated to elevated temperatures.

In figure 3 the resulting cooling capacities of the aforementioned fuels are compared. It was assumed that no fuel cooling is required below Mach 4 and that all cooling is done by the fuel above that speed. Only liquefied methane and hydrogen showed appreciable cooling capacity above Mach 5. Hydrogen is markedly the best fuel for cooling purposes, largely because it can be heated close to the limiting temperature of the cooled surfaces.

It should be pointed out that the Mach number at which the heat load exceeds the fuel sink capacity can be extended by flowing excess fuel into the combustor. This fuel-rich operation reduces the impulse, of course, but at a rate that decreases with increasing speeds.

## RANGE

It has thus been indicated that there exists no fundamental limit that precludes flight in the atmosphere to Mach numbers approaching and exceeding 10. This does not mean that flight at that speed is desirable. One obvious question is what ranges are attainable at these hypersonic speeds. Some of the considerations necessary to answering this question are shown in figure 4. The range equation,

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$$\text{Range} = IV \frac{L/D}{1-(V/V_s)^2} \ln \frac{1}{1-W_f/W_g}$$

consists of the terms, impulse, velocity, lift-drag ratio, centrifugal-force effect, and a log function of fuel- to initial-gross-weight ratio. All of these terms except velocity and centrifugal force decrease with increasing flight speed in the indicated manner. The net result is that range will maximize at some point in this speed range.

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Much of the material in the following papers will discuss how to attain the highest possible values of the terms over which there is some control: impulse, lift-drag ratio, and fuel- to initial-gross-weight ratio. In this regard it should be noted that the discontinuity in the variation of fuel- gross-weight ratio illustrates one method of maximizing this value at the start of cruise. That is, to provide a disposable booster as must be done in the case of the ramjet engine.

## SELF-BOOST

Since the ramjet engine requires at least some boost, differentiation is necessary between this engine type and those utilizing turbine-driven compressors with take-off capabilities. The distinction may be clarified with the aid of figure 5. At speeds much below Mach 1 the ramjet produces no useful thrust, whereas relatively low pressure ratio compressors are quite effective. In general, at low speeds, the higher the pressure ratio, the better the performance. As speed is increased, however, the higher the pressure ratio, the sooner the performance falls below that of the ramjet. The compressor and turbine are merely in the way at high speeds where most of the compression occurs in the air-induction system.

Because the self-boost capabilities of the turbine type engine are essential in some applications, paper 3 is devoted to discussing the various turbine cycles that may be utilized to drive the compressor.

## MATCHING

Among the many problems introduced by operation over a wide speed range, as required by self boost, is that of matching the air inlet and the jet exit nozzle to the air-handling capacity of the engine. This problem is illustrated in figure 6 where relative areas of an ideal inlet and exit are plotted as a function of flight Mach number for a hypothetical turbojet engine. The ideal areas are merely the areas of the capture stream tube and the discharged jet at ambient static pressure.

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The problem is simplified by considering the engine as approximating a fixed throttling device. The higher the flight speed, the more air can be forced through the engine. Conversely, if the inlet and nozzle are sized (as in the sketch of fig. 6) to capture and discharge the airflows at Mach 4, they are much too large at Mach 1.5. Unless the inlet is varied to bypass the excess air in a sophisticated manner at off-design speeds, large drags can result. The nozzle must also be adjusted to the discharge stream-tube area or suffer thrust penalties. At the same time, the adjustment must not incur large boattail drags.

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Although these curves are for a turbojet engine, they look much the same for a ramjet engine having a fixed combustor and nozzle throat. At the higher operating speeds of the ramjet, the matching problem becomes much more severe as indicated by the increasing rate of change of stream-tube area with Mach number. Nozzle-throat-area variation somewhat mitigates this problem by providing a degree of engine flexibility. Nevertheless, it is very difficult to make a good cruise engine provide much self-boost capability for the hypersonic ramjet.

## FUEL HEATING VALUE

The basic engine types and some of their inherent off-design problems having been introduced, it is of interest to return to design-point operation and the problem of maximizing the terms of the range equation over which some control is possible. When the impulse term is considered, the heating value of the fuel is certainly of paramount importance. In figure 7 the heating values of the more prominent fuels are shown. The superiority of hydrogen is clearly indicated by a heating value 70 percent greater than that of its nearest competitor, diborane. This fact, combined with its greatly superior cooling capacity, makes hydrogen extremely interesting as a fuel for long-range hypersonic flight. One of its disadvantages, low density, will be considered later.

## DISSOCIATION LOSSES

It is not at all certain that all the heating value of the fuels listed in figure 7 can be realized. The combination of high temperatures and moderate pressures in the combustion chamber at high Mach numbers results in dissociation of the fuel and air into many components. This dissociation absorbs energy and unless the components recombine into the products of combustion within the nozzle, the full heating value of the fuel is not realized.

The implications of this possibility are illustrated in figure 8 where thrust per unit airflow is plotted as a function of flight Mach number. The upper curve represents the thrust obtained with equilibrium

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expansion (full recombination) and thus represents full realization of the heating value. The lower curve, denoted frozen expansion, corresponds to the maximum loss due to dissociation. The difference between the two curves thus represents the loss in sensible enthalpy.

## RECOMBINATION



The possible losses clearly become very large at hypersonic speeds and whether or not equilibrium expansion occurs is a question of major import. Figure 9 illustrates this question with an example using hydrogen as a fuel. The various constituents at the entrance to the nozzle are listed along with that percent of the sensible enthalpy loss that is tied up in the particular constituent. Within the nozzle the temperature drops because of the expansion of the flow. As the temperature drops, the indicated reactions begin to take place recombining the many constituents into the two products of combustion. If all these reactions go to completion, there are no dissociation losses.

Unfortunately, the rates of all these reactions are not known. In particular, those involving hydrogen molecules and hydroxyl radicals are in doubt, and these chemical species contain 58 percent of the potential enthalpy loss due to dissociation. While research proceeds to establish these recombination rates, the hope is that the reactions will go nearly to completion in the large nozzles which will be of concern. Most of the calculations to be presented will thus assume equilibrium flow, although the effect of frozen composition will occasionally be illustrated.

## COMPONENT PERFORMANCE

Obtaining large values of impulse involves more than large heating values. High efficiencies must be attained in the inlet and the exit nozzle as illustrated in figure 10 along with some other interesting observations. It is immediately apparent from this figure that very high impulse levels relative to a rocket may be realized. This, of course, is necessary for sustained flight in the atmosphere but also indicates the potential of the ramjet as a booster.

Spotted on the curves for Mach 4 and 7 ramjets are the inlet kinetic energy efficiencies corresponding to the particular values of impulse and inlet pressure recovery (kinetic-energy efficiency $\eta_{KE}$ is the efficiency of the inlet in converting the free-stream kinetic energy into pressure within the engine). The highest indicated value of $\eta_{KE} = 0.97$ represents the best of current inlets and corresponds to realization of most of the available impulse. It is interesting to note that this value may be achieved with a much lower pressure recovery at Mach 7 than at Mach 4

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and that good values of impulse may be obtained with much lower values of pressure recovery. In general, it should be remembered that $\eta_{KE} = 0.95$ represents good inlet efficiency.

The fact that increasing pressure recovery from 0.35 to 0.70 does not result in correspondingly large increases in impulse should not be taken to mean that the attainment of high pressure recovery is not important in itself. Under some circumstances it can be vitally important since, for a given developed engine, doubling the recovery doubles the airflow through the engine and more than doubles the thrust. For lightweight engines designed to fit a particular mission, however, $\eta_{KE}$ is more indicative of the impulse and the range.

Also shown in figure 10 is the decrement from ideal impulse due to using an actual nozzle having a velocity coefficient of 0.97 in addition to being slightly underexpanded (this decrement is smaller at Mach 4). Refined nozzle design may regain up to half of this loss. The following paper on Inlets, Exits, and Cooling Problems discusses in more detail the problems of attaining efficient performance of these components.

## LIFT-DRAG RATIO

Efficient performance of the inlet and exit components must include low drag as a factor, since this influences another term of the range equation, L/D. Of course, L/D is more importantly influenced by other factors that are discussed in paper 4.

Shown in figure 11 is the variation of L/D with flight Mach number for currently efficient wing-body combinations. The problem is to obtain as good or better values of L/D with actual long-range configurations, with powerplants installed, and with sufficient fuselage volume to store the required quantities of fuel. That this may be difficult is better understood when one realizes that the powerplants become an increasingly large part of the total configuration with increasing speeds. Also, use of hydrogen as a fuel necessitates low-density fuselages, which are detrimental to the attainment of high values of both L/D and high values of fuel- to gross-weight ratio, the remaining term of the range equation to be considered.

## REMARKS

This paper constitutes a sketch of the basic ideas to be explored in more detail by papers 2, 3, 4, and 5. The requirement of a new engine for the ultimate in manned bombers with take-off capabilities will be considered. The requirement of a new technology for the hypersonic ramjet missile will also be considered. Here, is invisioned a "cooled" missile

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with all surfaces glowing red hot; a missile that contains hydrogen fuel in both a cold liquid and a hot gaseous form. As imposing as the attendant problems may seem, they certainly lie ahead if the ultimate capabilities of the type of weapon are to be realized.

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## ULTIMATE AIR BREATHING ENGINES REQUIREMENTS

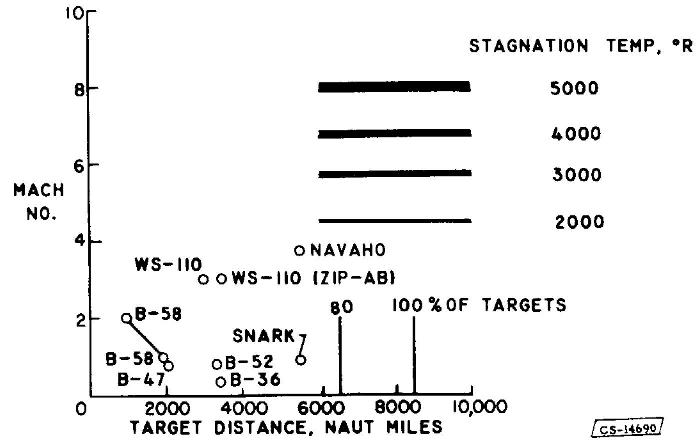


Figure 1

## COOLING IS REQUIRED

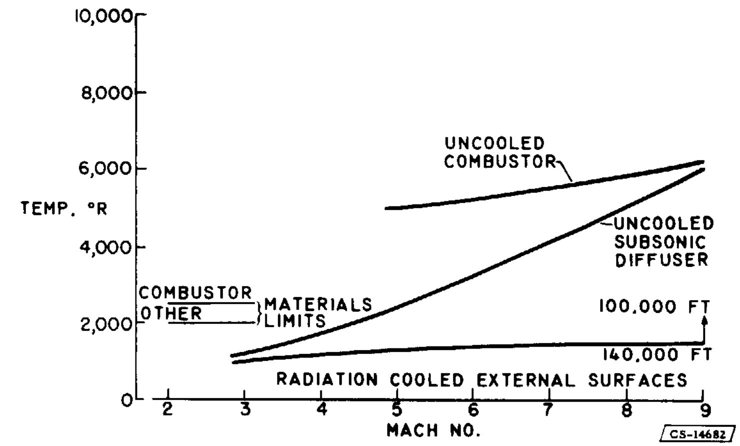


Figure 2

## FUELS AS COOLANTS

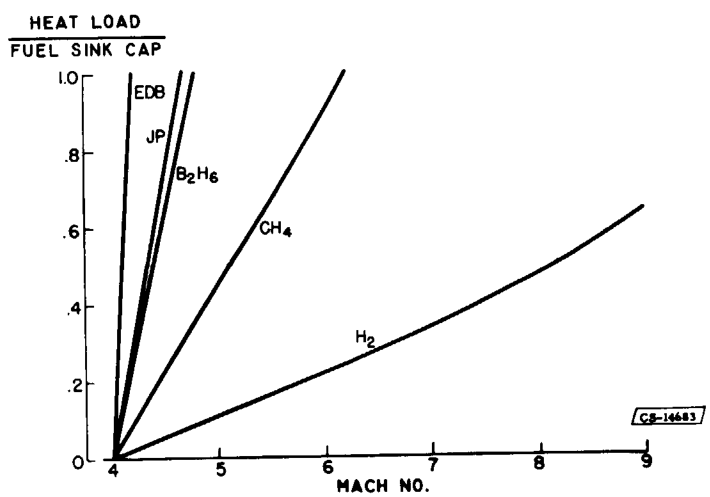


Figure 3

## RANGE CONSIDERATIONS

$$RANGE = IV \frac{L/D}{1-(V/V_s)^2} \ln \frac{1}{1-W_f/W_G}$$

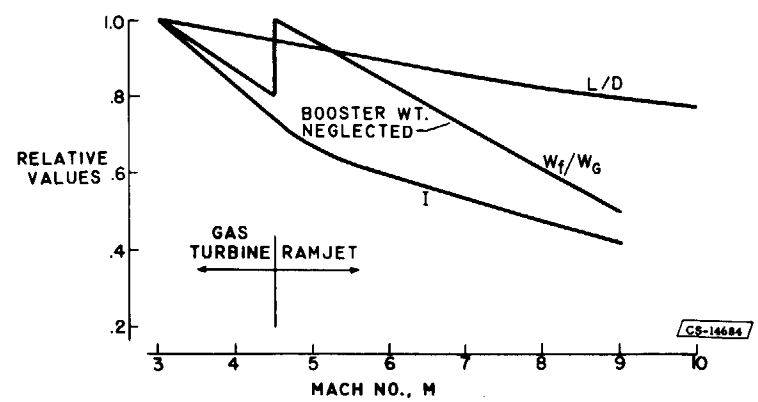


Figure 4

## SELF-BOOST PENALIZES HIGH-SPEED PERFORMANCE

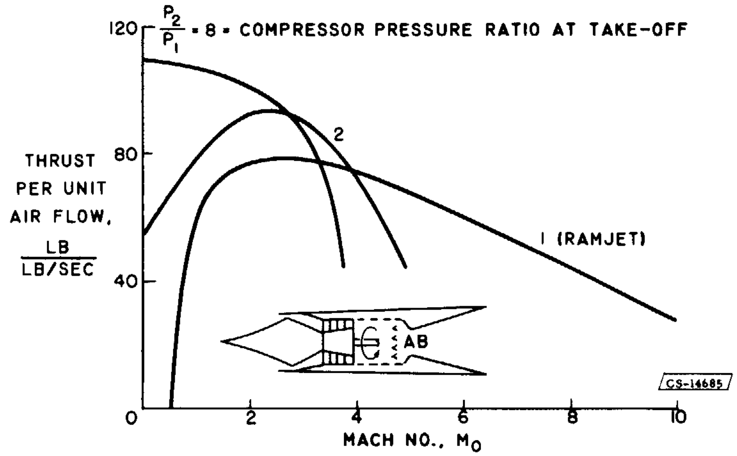


Figure 5

## TURBINE ENGINES HAVE MATCHING PROBLEMS

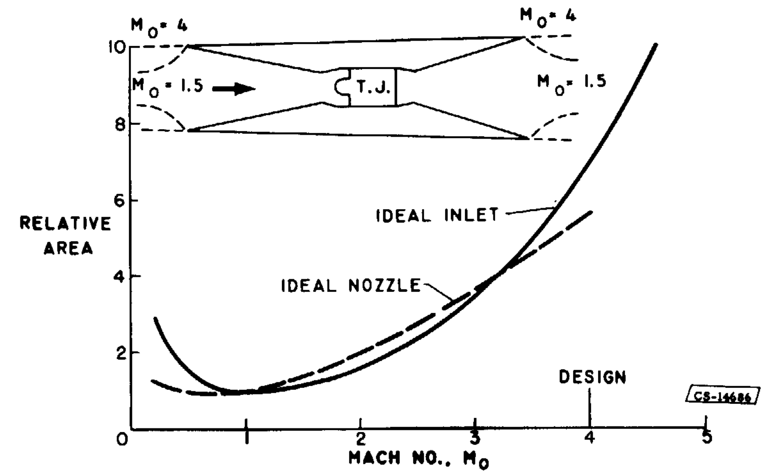


Figure 6

## FUEL HEATING VALUES

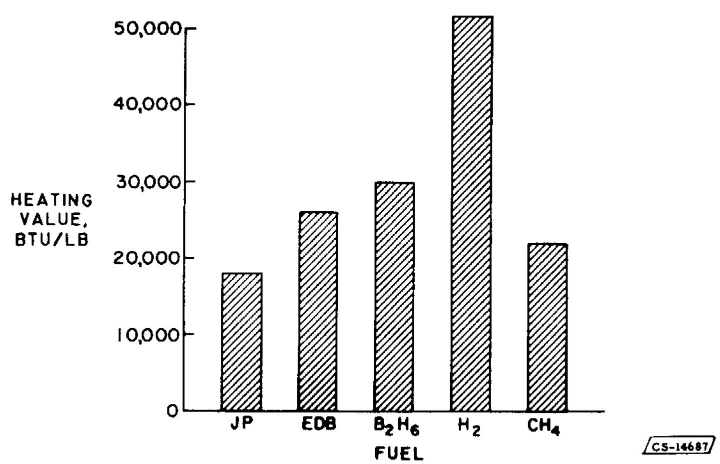


Figure 7

## DISSOCIATION LOSES THRUST

HYDROGEN FUEL

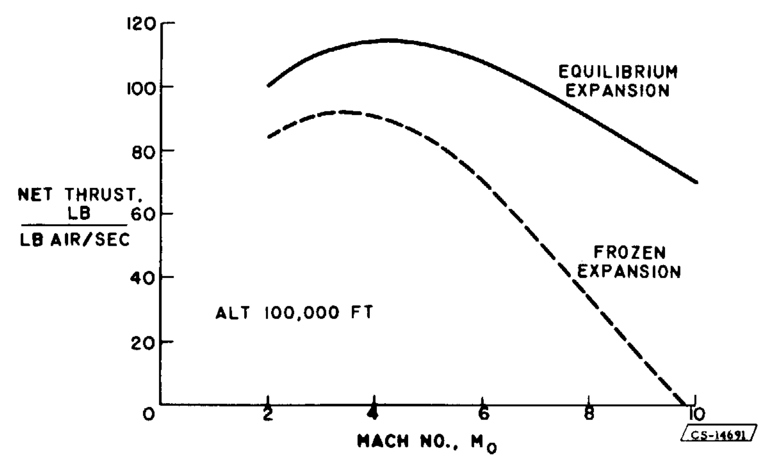


Figure 8

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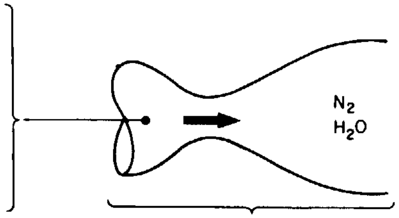
## RECOMBINATION IN NOZZLE

HYDROGEN FUEL

$M_0 = 7$ ALT, 100,000 FT STOICHIOMETRIC

| CONSTITUENT | % $h_{EQ}-h_{FROZ}$ |
|---|---|
| $N_2$ | — |
| $H_2O$ | — |
| O | 12.9 |
| H | 25.6 |
| $H_2$ | 35.5 |
| OH | 22.2 |
| NO | 2.3 |
| OTHERS | 1.5 |
| | 100.0 % |




$NO + H_2 \rightarrow \frac{1}{2} N_2 + H_2O$

$H_2 + O \rightarrow H_2O$

$2H \rightarrow H_2$

$\frac{1}{2} H_2 + OH \rightarrow H_2O$

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Figure 9

## HIGH IMPULSE ATTAINABLE

HYDROGEN FUEL (STOICHIOMETRIC)

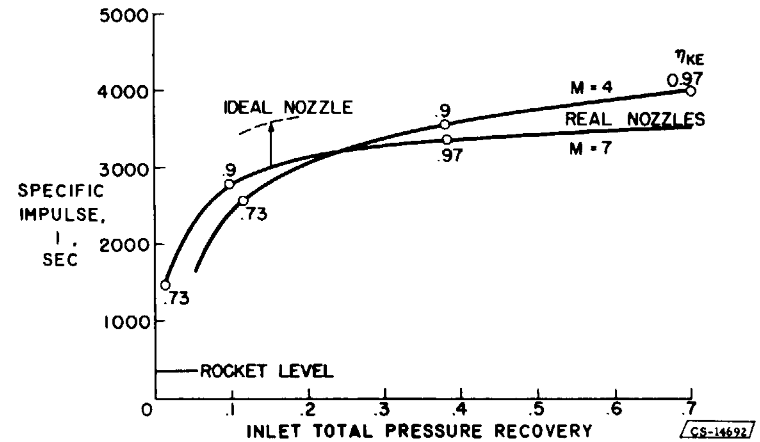


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Figure 10

## CONFIGURATIONS

RANGE ~ L/D

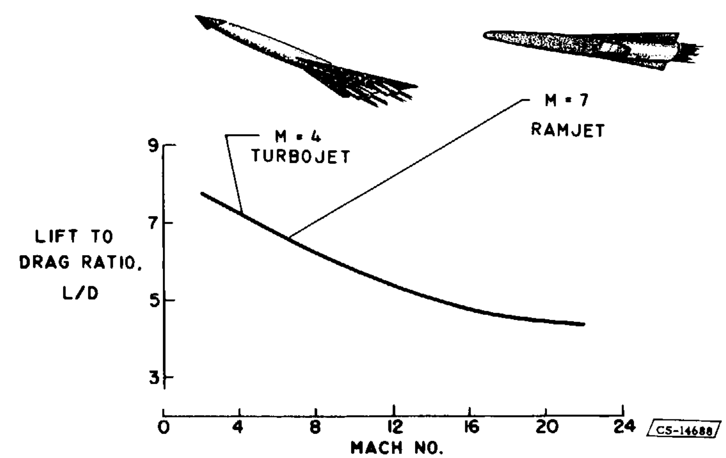


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Figure 11

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